

Tweeting from Earth, and 10k ft above ground level

Team 2 Project Technical Report to the 2024 Spaceport America Cup

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CARPO is a five inches high-power experimental rocket transitioning to a six inches airframe developed by Space Dragons that will contend in the Intercollegiate Rocket Engineering Competition (IREC) Spaceport America Cup in the 2024 edition within 10,000 ft AGL apogee with commercial-off-the-shelf (COTS) solid propulsion system. Just as important, performing a successful deployment of a payload capable of execute sensors to gather UV radiation data and broadcast real-time updates on X (Twitter), indicating whether it is safe or not to expose oneself to sunlight in the region based on the solar radiation index, taking care of a successfully rocket and payload recovery operation. The present project is an effort of several aeronautical engineering undergraduate students of their faculty advisors, who have designed, developed, and manufactured all the necessary elements for the mission, seeking to innovate in the Mexican experimental rocketry sector.

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Nomenclature

a	=	speed of sound (fps)
AGL	=	Above Ground Level
cm	=	centimeter
CAD		
CAE		
CFD		
COTS	=	Commercial-off-the-shelf
DOE	=	Design of Experiments
ft	=	feet
ft/s ²	=	aceleration
fps	=	velocity
g	=	grams
ICNIRP	=	International Commission on Non-Ionizing Radiation Protection
in	=	inches
IREC	=	Intercollegiate Rocket Engineering Competition
lb	=	pounds
M	=	Mach
m	=	meter
mm	=	millimetre
N	=	Newtons
Ns	=	Newton second
PBC	=	Printed Circuit Board
PLA	=	Polylactic Acid
PVC	=	Polyvinylchloride
QFD	=	Quality Funtion Deployment
S	=	surface
UNAQ	=	Universidad Aeronáutica en Querétaro
UNDP	=	United Nations Development Programme
WHO	=	World Health Organization
WHO	=	World Meteorological Organization
ρ	=	air density

I. Introduction

This project is developed by the Space Dragons Rocketry Team made up of thirty aeronautical undergraduate students from a variety of specialties such as manufacturing, mechanical design, electronics, and aircraft systems control, representing our university and our hometown, the State of Queretaro, Mexico. The project pursues to develop a COTS solid propulsion launch vehicle that under an aeronautical grade methodology can perform a successful launch and recovery after reaching an apogee of 10,000 ft above ground level (AGL). The aim is to reach an apogee altitude within 5% error margin stipulated by the competition along with dual deployment, including ejection of the payload and the corresponding recovery systems. The development of the project is a collaborative and integral work of the areas in which the team is divided, motivated by the aerospace development of our region and to contribute to the atmospheric sector of the country using UV sensors to broadcast the UV Index in real-time on social media platform.

II. System Architecture Overview

General Overview

Space Dragons Rocketry Team will compete in the Spaceport America Cup 2024 hosted by the ESRA and governed by the Intercollegiate Rocket Engineering Competition rules and regulations in the 10,000 ft AGL apogee with COTS solid propulsion category with a 75 mm White Lightning™ AeroTech M1850W-PS solid propulsion engine. The rocket is 10.9 ft height a lower a diameter of 5 in applying a transition to an upper diameter of 6 in and having

a 2.08 caliber stability will deploy a payload with UV Index application commanded by its on-board flight computer. The avionics bay comprises two COTS RRC3 dual-deployment altimeters, one serving as the primary deployment system and the other providing redundancy. It also houses a telemetry on-board computer, designed to transmit real-time flight data, which includes a microcontroller, an IMU, an altimeter, and a GPS unit. Additionally, the avionics bay is equipped with an independent Backup GPS to ensure continued geolocation in the event of a telemetry system failure.

Propulsion Subsystem
 Within **Error! Reference source not found.** the characteristics of the proposed M-type solid motors for the participant team launch are listed to quick check their general specifications a selected the adequate option.

Table 1. Main characteristics for solid motor selection. *Grouping critical information for engine selection according to performance and apogee AGL in line with the team's mission*

Parameter	M1350W-PS [1]	M685W-PS [2]	M1850W-PS [3]
Total Impulse	5,178.200 Ns	7,741.000 Ns	7,659.000 Ns
Average Thrust	1,356.700 N	657.000 N	1,914.750 N
Peak Thrust	1,765.500 N	1,517.000 N	2,446.000 N
Thrust Duration	3.800 s	11.500 s	4.000 s
Fabricant	AeroTech Motors	AeroTech Motors	AeroTech Motors
Propellant Weight	5.930 lb	9.200 lb	8.760 lb
Motor Weight	10.600 lb	15.150 lb	15.130 lb
Motor Diameter	2.965 inches (75 mm)	2.965 inches (75 mm)	2.965 inches (75 mm)

In this project, the team is planning to use the AeroTech M1850W-PS solid engine as a propulsive medium. It is projected to reach an estimated apogee of 9,963 ft justified beyond in this document based on the rocket geometry, weights, optimizations, and performance.

We did a test which is described later on the document with an AeroTech M685W-PS, were we reach an 8446 ft apogee; because of that we decided to do a comparative with the M1850W-PS to determine if it’s a better option.

Table 2: Main characteristics for new selection motor. *After test flight, the team decided to look for a motor alternative*

Parameter	M1850W-PS	M685W-PS
Theoretical Apogee	10,147.000 ft	9,725.000 ft
Maximum velocity	808.000 ft/s	666.300 ft/s
Velocity off rod	88 ft/s	65 ft/s
Thrust duration	11.000 s	4.000 s
Lift off to weight ratio	10.1	5.07

As a result of this comparative, we determined to use a M1850W-PS for this mission, also, as recommended in our feedback from 2nd report our lift of to weight ratio needs to be at least 6, with this modification this requirement is fulfil.

Aerostructures Subsystem

1. General Description

The structural and aerodynamic objective of this rocket comes on establishing a design that proposes the highest aerodynamic efficiency of the rocket and delivers adequate space for payload and flight computer section to ensure unobstructed dual deployment. Furthermore, thorough evaluation of the optimum weight-to-strength ratio of materials is sought. The project presents a transition section of diameters from 5 in to 6 in composed of three airframes, four fins, three centering rings, one inner tube, one nose cone, one motor bulkhead and one boat tail taking the role as engine retainer, as well. Materials were implemented such as fiberglass, carbon fiber, PLA filament, 2024 aluminium, among others, stretches a special focus on the application of appropriate manufacturing methods for each type of material used. The proposed design is based on the integration of the payload and the main recovery system in the upper airframe segment called *Forward Airframe A*.

The main objective of this integration is to provide more space and ensure clear deployment. Simultaneously, the flight computer will be in the "Forward Airframe B" section, which will be connected to Upper Fuselage A by a transition. The engine and fin section will be attached by an arrangement of two fuselages in the lower part of the rocket which contains the structure to support the engine, as well as a boat tail. The rocket has a total length of 10.9 ft and a total weight of 55 lb. The critical aerodynamic components of the rocket such as the nose cone, transition and fins have been analysed by ANSYS software to find the optimal shape and reduce the drag coefficient.

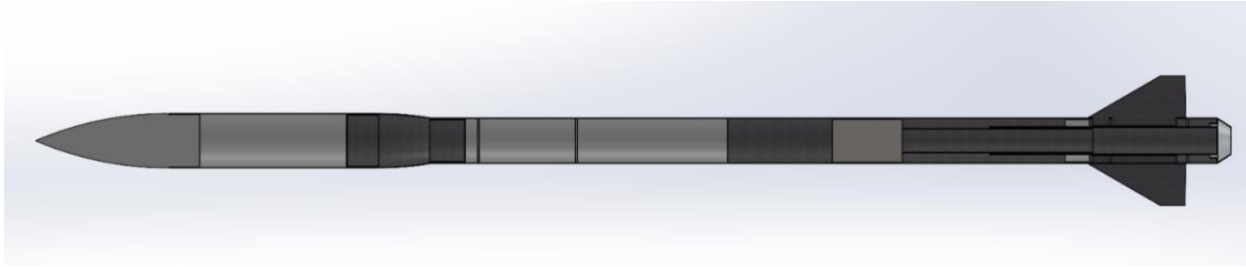


Figure 1: CARPO sectional view. CARPO is a modular design transition from 5 in airframe to 6 in permitting rapid assembly and disassembly of individual subsystem of individual subsystems.

1. Manufacture by molding

Since the team decided to manufacture our own composite rocket components based on CFD geometric optimizations, Space Dragons designed and manufactured 3D printed molds to keep the CAD shapes which were the optimal. Molds play an important role in the laminating process of composite materials since they can be helpful to create complex surfaces which can help us to increase the rocket's development and efficiency. These molds can provide us with a highly accuracy and consistent shape as the composite material will adhere correctly to its form, this guarantees the cured product to have the desired dimensional specifications and tolerances.

2. Nose Cone Analysis

The analysis focuses on optimizing the nose cone for a rocket based on drag performance at speeds of 0.6 to 0.9 Mach. Four geometries (Haack Series, VonKarman Series, parabolic nose cone, and exponential series with $n=0.5$) showed promising performance. The Haack series, particularly LV-Haack and LD-Haack, exhibited minimal drag.

Haack series nose cones, unlike other types, are not designed based on geometric figures but on curves derived from the Bernoulli-Euler beam theory. There are two characteristic types:

1. LV-Haack: Resistance is minimized for a given length and volume.
2. LD-Haack (Von Karman): Resistance is minimized for a given length and diameter.

Equation (1) is representing such VonKarman Series curve profile:

$$y = \frac{R \sqrt{\theta - \frac{\sin(2\theta)}{2} + C \sin^3(\theta)}}{\sqrt{\pi}}, \theta = \cos^{-1} \left(1 - \frac{2x}{L} \right) \quad (1)$$

Where C takes the value for each case presenting below:

$$C_{LV} = \frac{1}{3} y C_{LD} = 0$$

Excel was used to obtain geometric values and plot profiles for Haack series configurations. ANSYS Workbench was employed for further analysis. A 2D axisymmetric analysis was conducted, resulting in minimum drag values of 0.256 for Haack LV and 0.263 for Haack LD.

3. Nose Cone

The shape of the nose cone performs a fundamental role in reducing drag and hence improving the aerodynamic efficiency of the rocket. The CFD optimizations are presented in Section 3: Nose Cone Analysis where has been proposed a Series Haack-Shaped nose cone model with a 6 in diameter and length of 14.96 in The manufacturing process is by moulding of carbon fiber impregnation method using a mold made from PLA filament, 4-layer walls each one of 0.007 in thickness. This choice is based on an optimum balance between impression time and structural integrity, supported by data from a study carried out at the University of La Laguna [4]. In this regard, this method represents the sole approach that allows the team to precisely determine the optimal nose cone geometry for the best performance, thereby eliminating reliance on commercially available options.



Figure 2: 3D Printed Half Nose Cone Mold. To ensure ANSYS optimization nose cone points, it has been decided to manufacture our own nose cone

The entire manufacturing process is carried out using two molds that has previously been sanded to achieve the smoother surface possible. Subsequently, a layer of epoxy resin is applied and finished with 1000 grit sandpaper. Then, before applying the carbon fiber previously impregnated with epoxy resin, the molds are prepared using wax and a plastic release agent to make easier the demoulding process. In one mold, the excess fiber that protrudes from it will be cut right at the border, while in the second mold the carbon fiber will be placed and a glassfiber excess of 1.5 in which will be left on the edges to be able to attach both fiber parts together. A resin-fiber ratio of 78:22 was obtained throughout a manual layup, as detailed in Annex 1, ensuring a high standard relation to ensure structural properties. The nose cone is left to cure for 24 hours, and in this way, it obtained a thickness of 0,07 in.

For the nose cone tip, AISI 2024 aluminium has been selected to manufacture because offers unique characteristics just as increased resistance to stresses, withstand extreme thermal conditions, light weight, and machinability that makes the most suitable choice for CNC work and to preserve the precision of the geometry.



Figure 4: Carbon Fiber Wet-Preg Procedure for Nose Cone. Composite lay up of the first half of the nose cone

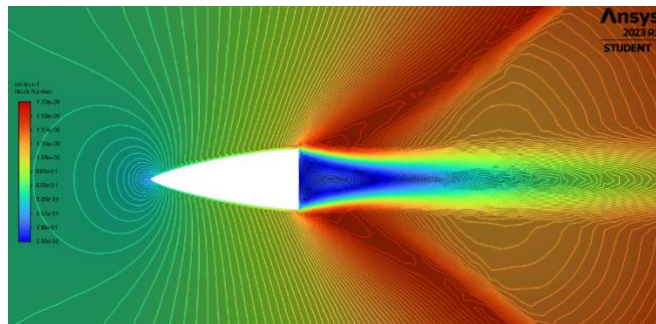


Figure 3: ANSYS Fluent Representation. Velocity analysis of the nose cone made it in ANSYS Fluent

4. Upper Airframe A and B

Three airframes will be used for this project. The forward airframe A will have an external diameter of 6 in, being that the payload will be included inside. In addition, the forward airframe B with 5 in diameter is used, considering that it will not be deployed, but lay up the flight computer. Additionally, as the payload carrier, it is necessary to avoid the creation of interference in the communication signals as would be generated by implementing carbon fiber. Therefore, the manufacturing process was through fiber rolling a pre-impregnated fiberglass with epoxy resin using a steel cylinder tube as

mould which was prepared with wax and plastic release. This method offers the opportunity to remove excess epoxy resin to reduce the weight of the airframe. Both fuselages were constructed of fiberglass with a fiber-to-resin ratio of 5:13 for both, achieving a final thickness of 0,12 in and after the rolling process, both airframes were sanded increasing gradually the sandpaper from 100 to 3000 grade.

5. *Transition analysis.*

ANSYS analysis aimed to optimize the rocket's transitions, aiming to reduce the drag coefficient (CD), to match that of the nose cone at 0.258 CD. Using SolidWorks, the rocket was modeled with various transition configurations, from Haack LD to tangent curves. Parameters such as dimensions, shape, and materials were defined for then, conduct numerical simulations in ANSYS Fluent's CFD module iterating between a minimum and maximum range of geometric parameters. The focus was on identifying configurations with a notable reduction in CD. The manufacture follows the same procedure as the nose cone.

6. *Diametral Transition and Couplings*

The execution of a transition in the rocket combines design versatility with ease of assembly in order to connect two airframes with different diameters in the most non-aggressive and aerodynamic way between these sections ensuring an adequate airflow during flight. These components were manufactured from fiberglass to guarantee its resistance to aerodynamic loads. Geometrically, the transition had a length of 5.906 in, an initial diameter of 5 in and a rear diameter of 6 in. In addition, the rocket is having two couplings along its length. The manufacture follows the same procedure as the nose cone. The transition was joined to the forward upper airframe A and B by structural fasteners. This preference is due to the ease of assembly and disassembly offered by bolts, allowing fast disassembly and assembly. Four countersunk ¼"-20 stainless screws are used in order to not significantly increase the weight, but at the same time to provide good fastening into the transition. The carbon fiber coupling will join the upper airframe B to the lower airframe A by using four shear pins. Every single coupler was made by the team, which the major challenge was ensuring a high-precise tolerancing in order to strictly prioritize the elimination of any gaps or clearances that could induce buckling between airframes.

7. *Aft Airframe*

The Aft Airframe with a 5 inches inner diameter and 41.33 inches of length has the function of being the motor carrier for the rocket, having inside it an internal carbon fiber tube also of 3 inches, length of 35.43 inches and a thickness of 0,07 in joined to three aluminum centering rings with a thickness of 0,5 in. Additionally, it was added an aluminium bulkhead that helps to transmit the thrust generated by the motor to the Aft Airframe and to the rest of the rocket. Similarly, the fins are attached to the fuselage, all the above-mentioned components are joined by using epoxy resin and Cab-O-Sil in a proportion of 1% of the total weight of the mixture.

8. *Fin Analysis*

One of the improvements the team undertook this year was an exhaustive CFD and CAE analysis conducted using a finite element method software to optimize the design, especially of fins. The main objective was to reduce the rocket's drag coefficient (CD) by modifying various fin measures and observing their impact on aerodynamic performance.

The optimization of the rocket fins through Ansys analysis proved to be effective in reducing aerodynamic resistance. Modifying measurements such as the area and length of the fins allowed us to find a more efficient design in terms of CD. Following comprehensive analysis, trapezoidal fins have been identified as the optimal choice for this rocket. Their advantages include ease of manufacture, providing directional stability during flight, and minimizing aerodynamic drag, thereby facilitating higher altitudes and speeds. Although trapezoidal fins may have limitations in maneuverability for rockets requiring abrupt changes in direction, they align well with the project's goals.

Through ANSYS Workbench simulations, the team took two fins geometry to compare their benefits and simulate to been validated, considering crucial factors such as altitude, speed, and material properties.

After conducting multiple simulations and optimizations, it was observed a reduction in the coefficient of drag (CD) of the experimental rocket between the geometry 1 and 2 of the fin. A model of the rocket with its fins was generated in the Ansys and SolidWorks environment, utilizing the original measurements and proportions as a starting point. Subsequently, flow simulations were conducted over the rocket model using expected flight conditions, recording drag coefficient (CD) values for subsequent analysis. Several iterations were performed by modifying the fin measurements, with each design evaluated through simulations to determine its impact on the CD.

Ultimately, the design that exhibited the greatest reduction in CD without compromising other aspects of rocket performance, such as stability and maneuverability.

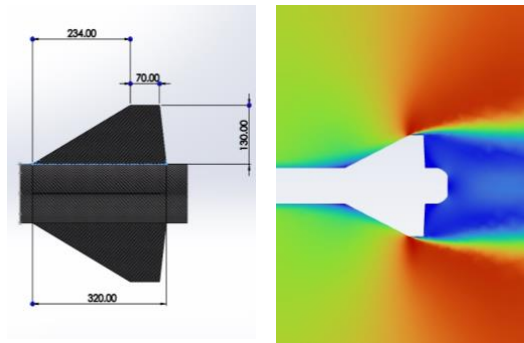


Figure 5: Geometry 1 of Finset ANSYS comparison.
3D fin representation and fluid analysis.

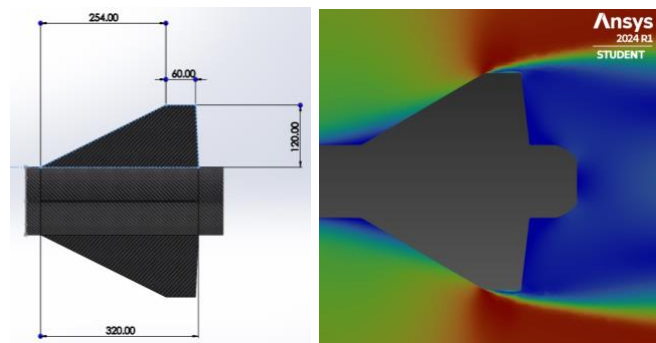


Figure 6: Geometry 2 of Finset ANSYS comparison.
3D fin representation and fluid analysis.

With geometry 1 (2.36 in), was obtained a drag coefficient of 0.3921, while in the second geometry (2.75 in), we obtained a CD of 0.4046. Therefore, we determined that the geometry to be used for our rocket will be the first. Modifying measurements such as the area and length of the fins allowed us to find a more efficient design in terms of CD. These results support the importance of conducting detailed aerodynamic analysis in the design of spacecraft and aircraft.

9. Fins attachment

After the fins were analyzed, they were attached to the rocket which were manufactured using a sandwich structure where 4 layers of carbon fiber arranged at 0°, 45° and 90° covered one side of a layer of glass fiber mat, then the layers were closed by placing another 4 layers of carbon fiber in the same directions, for the attachment, first the slots were made to the lower carbon fiber fuselage of the length and thickness of the fin root. Then, the fins were attached to the inner tube and two centering rings using epoxy glue. Once the fins were fixed and 90° perpendicular to the tube, the fillets were made using the same epoxy glue.

a. Fin flutter justification

The following formula obtained from Apogee Components was used as a reference for the theoretical analysis of the fin flutter and the data obtained from OpenRocket and Rasaero II simulations, predicted atmospheric conditions in New Mexico, as well as humidity and temperature, were used. On the other hand, an important data to consider is the Young's Modulus that has helped to calculate the flutter. To obtain this data, specimens of the same material were made and by means of a destructive test of the material it was possible to obtain. With the data we were able to obtain a flutter speed of 1649 fps which compared to our simulated maximum speed of 810.9 fps we can conclude that our flutter is within acceptable range of safety.

b. Modal Analysis

Flutter is an unstable, self-excited structural oscillation at a natural frequency where the structure extracts energy from the airstream which can result in a catastrophic failure due to a structural resonance. Most of times this vibration, in terms of flutter, is the combination of two or more modal forms (vibration pattern of the resonance).

In this analysis, the modal analysis system form ANSYS workbench will be used to determine the fin's natural frequencies. First, modal modulus will be dragged to the project schematic to change the material's properties as we need carbon fiber's density, Young's modulus, and Poisson's ratio as seen in figure 13.

Properties of Outline Row 3: Structural Steel		
A	B	C
Property	Value	Unit
1		
2	Material Field Variables	Table
3	Density	1750
4	Isotropic Secant Coefficient of Thermal Expansion	lg m^-3
5	Isotropic Elasticity	
6	Derive from	Young's Modulus and Poisson...
7	Young's Modulus	3.6187E+11
8	Poisson's Ratio	0.27
9	Bulk Modulus	2.6222E+11
10	Shear Modulus	1.4247E+11
11		

Figure 7: Material boundary properties.

The fin's geometry was imported in a previous Parasolid model, then a properly mesh was generated to make it easier for the program to solve this analysis.

A range from 1 Hz to 1 000 000 Hz was suggested to find several natural frequencies our structure is oscillating and 10 modal forms to be found were set. A fixed support where the fin is attached to the fuselage was established as the boundary condition. In that way, the analysis was solved giving us ten established total deformation contours.

ccFinally, below it shows the natural frequencies where this structure is unstable, and it freely oscillates causing structural damage.

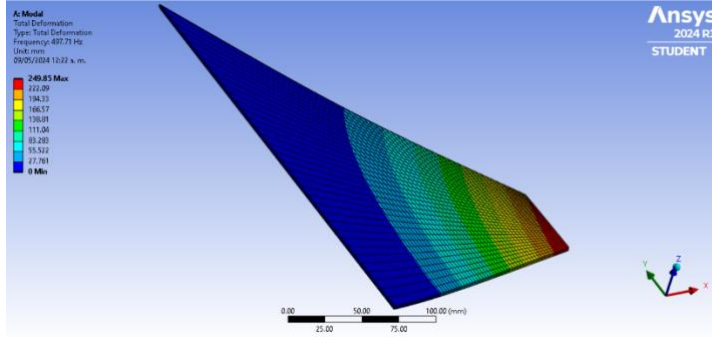


Figure 8. Sample of the analysis for the 1st modal form.

Mode	Frequency [Hz]
1.	497.71
2.	1276.1
3.	2433.8
4.	2749.8
5.	4159.2
6.	4872.9
7.	6040.9
8.	7072.9
9.	7522.6
10.	8485.

Figure 9. Natural frequencies of the first 10 modal forms in fins

Rockets need to rely on thoughtful construction and air to damp out any vibrational energy in the fins. Air is very efficient at reducing the amplitude of the vibration while the rocket remains under the flutter velocity. However, once the flutter velocity is exceeded the air will exponentially amplify the oscillations and rapidly increase the energy in the fin to the point of destruction.

$$V_f = a \sqrt{\frac{GE}{\frac{39.3 * A^3}{\left(\frac{t}{c}\right)^3 * (A + 2)} * \left(\frac{\lambda + 1}{2}\right) * \left(\frac{p}{p_0}\right)}}$$

- V_f represents the computed velocity at which fin flutter occurs.
- 'a' denotes the speed of sound at the altitude where the rocket reaches its maximum velocity above sea level.
- GE stands for the shear modulus of the fin material, measured in pressure units.
- A signifies the fin's aspect ratio, which is determined by the square of the fin's semi-span length or height divided by the fin area, and is dimensionless.
- t/c represents the fin's thickness ratio, calculated as the ratio of fin thickness to root chord length, and is dimensionless.
- λ (lambda) indicates the fin's taper ratio, defined as the ratio of tip chord length to root chord length, and is dimensionless.
- p/p_0 is the ratio of air pressure at the altitude where the speed of sound was determined to air pressure at sea level, and is dimensionless.
- The constant "39.3," whose value varies depending on the fin's geometry, is expressed in pressure units and is computed as explained below.

For the speed of sound in air, denoted by 'a', relies solely on temperature due to the cancellation of altitude-related effects resulting from decreasing density and pressure, assuming air behaves as an ideal gas. In this scenario, the speed of sound is determined by the following equations:

$$a\left(\frac{ft}{s}\right) = 49.03 * (sqrt(459.7 + TF))$$

To accurately calculate the speed of sound, it's essential to determine the air temperature at the altitude where the rocket reaches its maximum velocity, typically just before motor burnout as indicated by typical engine thrust curves. This altitude,

known as Above Ground Level (AGL), that we obtained from OpenRocket. Once the AGL altitude is determined, it should be added to the launch site altitude to obtain the altitude above sea level. In this case the data from the launch site.

$$TF = 30.43 - (0.00356 * altitude)$$

For the shear modulus (GE) of material we use the Young Modulus obtained from our strain stress and calculated with the following formula. Where v is taken as the Young Modulus.

$$GE = \frac{E}{2 * (1 + v)}$$

For the fin area the next equation helps to obtain it.

$$Area = \frac{Height * pi * \frac{root\ chord\ length}{2}}{2}$$

To determine the air pressure at a given altitude, one can employ the temperature previously computed in any of the subsequent equations.

$$P(psi) = 14.696 * \left(\frac{TF + 459.7}{518.7} \right)^{5.256}$$

The denominator constant (DN), represented by "39.3" in Martin's equation, is notable for having pressure units. Its calculation proceeds as follows:

$$DN = \frac{24 * \epsilon * K * p0}{pi}$$

The value of 24 is a dimensionless product of the whole number constants found in Martin's derivation.

Epsilon (ε) represents the distance of the fin center of mass behind fin quarter-chord. Given by the next equation.

$$\epsilon = \frac{Cx}{RC} - .25$$

Where Cx represents the axial distance from the front of the fin

Kappa (K) is the ratio of specific heats for air.

After obtaining every data from the equations we used Excel software where we obtained a flutter velocity of 1649.78 fps

Data	#	Units			
Root Chord (Cr)	12.598	in			
Tip Chord (Ct)	2.36	in			
Semi Span (b)	5.512	in			
Sweep Lenght	10	in			
Thickness (t)	0.12	in			
Pressure	9.33897009	psi			
P0	14.696	PSI			
a (maximum rocket speed)	715.223	ft/s	Vf	1649.78322	ft/s
Poisson Ratio (v)	0.27				
Shear Modulus (G)	1872430	PSI			
Wing Area (S)	41.224248	in^2			
Aspect Ratio (AR)	0.73699692				
Ct/Cr (Lambda) (Tapper Ratio)	0.18733132				
Temperature	30.43	°F			
Speed of sound (a)	715.223	ft/s			
p/p0	0.63547701				
t/cr	0.00952532				
Cx	8.18270009				
TF	16.132684				
DN (Denominator Constant)	62.7957316				
e (epsilon)	0.39952374				
k (Kappa)	1.4				

Graphic 1. Boundary conditions for flutter calculation

Figure 18.- Excel Fin Flutter Calculations

c. Manufacturing and tip-to-tip

The calculations made previously had already contemplated a tip-to-tip for the fins, so it was done by applying 3 layers of carbon fiber impregnated with epoxy resin from the tip to tip in each of the 4 joints of the fins and then a vacuum was made to that section to remove excess resin that could be left in that section.



Figure 12: Tip-to-tip vacuum. After a quasi-isotropic arrangement, vacuum was applied for absorb the resin excess and get a 80:20 ratio



Figure 11: Manufacture of fins. The four fins were created in the plates, cut and assembled in the body tube

10. Boat-Tail and motor retainer

An analysis was conducted using Ansys software to observe the behaviour of Cd with and without boat tail. The main objective was to reduce the rocket's drag coefficient (Cd) by comparing both results, checking the behaviour of the boundary layer on the objective zone of study. A Haack LD curve was used to create the geometry of the boat tail. This was established because the values provided were the diameter and the large of the transition, thus the specified curve produced less drag than the Haack LV., a boat tail is proposed to facilitate smoother and more efficient airflow at the rocket bottom, which decreases the boundary layer separation. The boat tail with a fore diameter of 5.16 in, aft diameter of 2.99 in and a length of 1.575 in is aim achieved higher altitudes with the same amount of solid propulsion. The boat tail was CNC machined and takes the role to be at the same time the motor retainer as it is designed with a screw mechanism for easy installation.

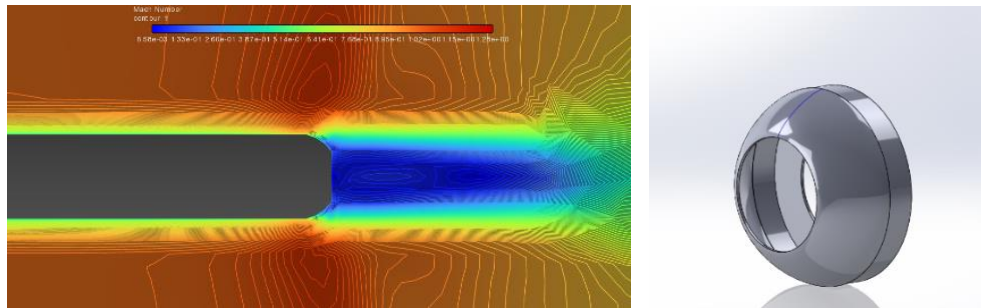


Figura 13: Boat tail and retainer redender and CFD simulation

11. Structural analysis

Determining the correct number of layers of composite material for an experimental rocket is paramount for various reasons. Firstly, it ensures the structural integrity of the rocket, safeguarding against potential failures during launch and flight. Secondly, it enables weight optimization, preventing unnecessary additions that could hinder performance and payload capacity. Moreover, it contributes to cost efficiency by minimizing material waste and fabrication expenses. Additionally, the right number of layers influences the rocket's overall performance, affecting aerodynamics, stability, and ultimately, mission success. Finally, prioritizing the correct number of layers enhances safety, mitigating the risk of catastrophic incidents that could jeopardize both the rocket and its surroundings. Therefore, meticulous consideration and determination of the optimal number of layers are essential in the design and construction of experimental rockets.

The first part of the structural analysis consisted in the realization of composite material specimens, to characterize the physical properties of the carbon fiber and epoxy resin that we will use in the manufacturing. At Space Dragons we have always worked with composite material such as carbon fiber and fiberglass giving a total of 4 layers on any laminated part, including fuselages, fins and nosecones. However, for Spaceport 2024 we have decided to implement in the technical report the stress simulation on each of the components of our rocket. The resin applied for the specimens was polyform epoxy, with a concentration of 80:20 according to the aeronautical normative recommendations, in addition, the technique used for the manufacture of the specimens was the wet preg, then applying vacuum for a period of 24 hours for a uniform curing, smooth finish and absence of bubbles.

12. Deflection tests

To determine the correct number of layers of composite material in our rocket, we carried out a deflection test on specimens with different thicknesses (4, 6 and 8 layers of material). The bending test that was carried out was specifically three-point, the process carried out follows specific rules of the ASTM D 7264 standard, providing a structured approach to evaluate the mechanical properties of these materials. We use the Pasco - ME-8244 machine precisely to guarantee reliable and comparable results. Three-point bending test stands as a key method to understand how these materials behave under load. The ASTM D 7264 standard establishes specific guidelines for performing the three-point bending test on composite materials, focusing especially on those reinforced with continuous fibers. Guaranteeing the consistency and validity of the results.

The test execution is performed using the Pasco - ME-8244 machine, selected for its ability to be configured to ASTM D 7264 standards. This machine provides a controlled and precise platform for the load application, ensuring that the test conditions test is reproducible and comparable.

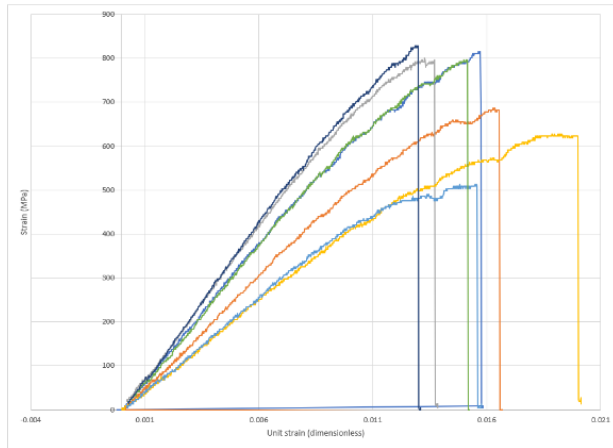


Figure 15: Graphical results of the strain test for 4 layers of fiber

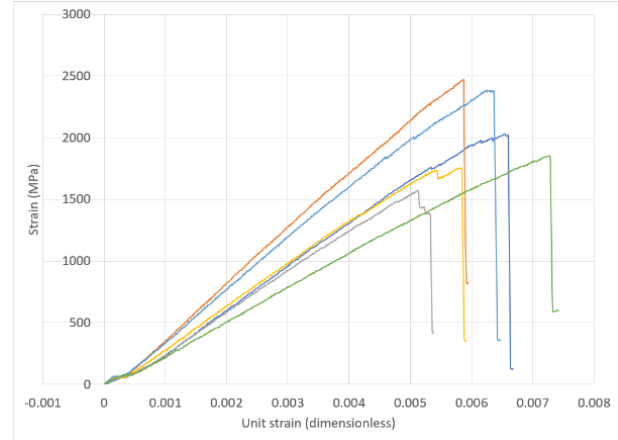


Figure 14: Graphical results of the strain test for 6 layers of fiber

13. Loads analysis

Experimental rockets are commonly vulnerable to stresses resulting from bending. In this analysis, aerodynamic flight loads will be determined as this can provide with a general idea of the rocket's behavior during flight. Most significant bending loads acting on an experimental rocket in flight are generated due to sudden wind gusts or a result of dynamic imbalance. Gusts, acting horizontally, generate normal forces on the rocket body. The nominal flight path will generate normal forces acting on the nosecone, fins, transition, and boat tail. In order for these forces to be developed, the inertia of

Table 3: Conditions to Normal Force Calculation

Component	Normal forces acting	Location of the forces
Nose cone	255.987 N	177 mm
Transition	-90.62 N	1252.28 mm
Boat Tail	-109.7 N	3298.2 mm
Finset	829 N	3068.95 mm

the rocket reacts to these. The resulting forces generate a bending moment applied to the rocket body. All these mentioned forces are applied at each component's center of pressure. The normal force acting in each component will be developed, these forces are given by the maximum dynamic pressure, a reference area, angle of attack and the normal force coefficient. It can be seen in the table 4 some relevant data, boundary conditions, and geometric parameters which will help with this analysis. With Annex 1, there is the math detailed explained and the boundary conditions used.

Equation (2) is used to calculate the determined drag force

$$F_D = \frac{1}{2} \times \rho \times v^2 \times A \times C_D \quad (2)$$

$$F_D = \frac{1}{2} (1.01) (186.536)^2 (0.020867) (0.398)$$

$$F_D = 145.935 \text{ N}$$

III. Recovery Subsystem

The mechanism of ejection system known as dual deployment [5] is discussed, referring to the release of two parachutes at specific times during the descent phase of a rocket. In the first instance, the drogue parachute is deployed at apogee around 9,950 ft AGL. Then, the main parachute is released at an altitude of approximately 1,000 ft AGL. For this event, an 5.6 grams of black powder for main charge is contained in cylinders covered. A redundant charge with 20% increase is added to ensure that the parachutes are ejected with a back-up system. The amount of powder is proposed in accordance with the SAC 2023 team participation. Conversely, the team have conducted the corresponding tests to confirm the amount of powder.

14. Black powder calculations

We part from the pressure formula and by experimentation we use 10 psi for pressure.

$$\text{Pressure (psi)} = \frac{\text{Force (lbf)}}{\text{Area (in}^2\text{)}}$$

Then we need the volume of the airframes both cylinder and transition

$$\text{Volume (in}^3\text{)} = \frac{\pi * \text{diameter (in)}^2 * \text{lenght (in)}}{4}$$

Next assuming the pressure of gas expands uniformly and instantly throughout the volume of the airframe.

$$W = \frac{PV}{R_c T} \rightarrow m = \frac{PV}{R_c T g}$$

Finally, we use the next equitation

$$\text{mass (g)} = \frac{454 \text{ g}}{1 \text{ lbm}} * \frac{\text{Pressure (psi)} * \text{Volume (in}^3\text{)}}{R_c \left(\frac{\text{ft} - \text{lbf}}{\text{lbm R}} \right) \text{Temperature (R)} \text{ gravity acceleration } \left(\frac{\text{ft}}{\text{s}^2} \right)}$$

15. Ground Test Results

For the ground test, it were tested the nosecone deployment and the drogue parachute deployment; for the main the BP used were 5 g for the main charge and 6 g for redundant, meanwhile the drogue used 3 g for main charge and 3.5 g for



Figure 17. Ground ejection test for drogue recovery system. The team used 4 gr for principal Black Powder and 3 gr for redundant.



Figure 16. Ground ejection test for main recovery system. The team used 5.6 gr for principal Black Powder and 4.5 gr for redundant.

redundant; the redundant charge will detonate 1 second after the main charge and as shown, the redundant charge has a 20% weight increase to make sure the deployment is successful.

16. Descent rate calculation

On the other hand, the diameter of the parachute is crucial to establish the level of descent rate is used to determine the optimal parachute diameter:

$$S = \left[\frac{2 \cdot g \cdot m}{\rho \cdot C_d \cdot v^2} \right] \quad (3)$$

In this project the drogue parachute is a commercial 29 in diameter from Animal Motors [6] which offers a calculated descent rate of 88 ft/s. Subsequently, the main parachute of 72 in diameter, which provides an estimated descent rate of 34.624 ft/s, will be ejected. The shock cord is made of nylon from Apogee [7] model 300 which will withstand a minimum of 890 N and the force is within the operating range of the shock cord [7] from 2.2 kN to 8.8 kN. Additionally, this shock cord model has already been tested and used within different projects of the team including *Xitzin* in the last Spaceport America Cup 2023 competition. For payload, a nylon fabric parachute with a spherical geometry of 36 in diameter will be used as it offers a descent speed of 30 ft/s, sufficient to avoid a displacement of more than 4.5 km radially.

IV. Payload Subsystems

The mission of the CubeSat payload is to determine and disseminate healthy sun exposure times and schedules for the general population by surveying and analysing the intensity of UV Index radiation in the environment. It has been decided to take action in this health sector on UV rays because climate change and damage to the ozone layer have caused more harmful UV rays from the sun and reach the Earth's surface. As a result, the dangers to humans from excessive exposure to UV radiation have increased, including damage to the skin, eyes and general health [8]. In addition, large amounts of UV radiation also affect other living things such as plants and aquatic animals, altering their natural processes [9]. The mission of this project is to develop and implement a new approach to the study of UV radiation.

A. CubeSat

This technical report presents the viability of the CubeSat payload project in terms of mission complexity, access to mission information, knowledge of handling and using electronic components for mission development, knowledge of designing structural components for the CubeSat casing, and obtaining the required components and materials. The team have developed the software and manufacture the 3U CubeSat in order to archive:

- Census of UV intensity and determination of solar UV index for the organisations WHO, WMO, UNDP and ICNIRP.
- Determination of healthy sun exposure times and schedules through the analysis of GPS data, solar UV index and health indicators established by WHO.
- Generation and publication of informative content to promote protection against UV radiation in the general population on social network X (Twitter).
- In-flight video broadcast as eye-catching content to achieve the above objective.

In order to fulfill the SAC 2024 standards and qualify as a functional payload, the payload is a 3U CubeSat [12] with 2024 aluminium armours. The total weight of the payload is 8.8 lb, which is in line with Spaceport Cup CubeSat standards [10]. The use of aluminium is preferred due to its malleability and paramagnetic properties, which allow for a rigid, lightweight structure with low radio frequency data transmission interference rates. Aluminium 2024 is a suitable material for the manufacture of CubeSats due to its low density and corrosion resistance. This material significantly contributes to weight reduction, structural integrity, and durability hazard environment, essential for mission success.

Below are listed the main electronic components to implement and a short description:

- Raspberry PI 4 Processor: Processing for image analysis
- Raspberry PI NoIR Camera V2: camera without infrared light filter for image capture
- 64GB SD memory: storage image files
- GPS NEO 6: obtaining payload location coordinates for payload recovery
- Xbee PRO S3B transmitter module: real-time transmission of payload GPS location with a range of more than 9.6 km distance

- Xbee USB adapter: Xbee PRO S3B communication interface and Raspberry PI 4 processor.
- SMA antenna: real-time data reception between payload to ground station
- 9V battery: power supply for the electronic systems
- Power Bank with 5V/3A output: reduces the number of components on the power stage and provides sufficient power to the payload subsystems for several hours

B. Design and manufacturing

A 3D design was created for the payload, defining the measurements, positions, tolerances of each electronic component on board, and the structural requirements for the armour casing (See **Error! Reference source not found.**).

The payload case is manufactured from aluminium plates approximately 0.4 in. thick. It consists of four side covers, a top cover, a bottom cover and three internal separator plates. Each piece was adjusted to fit perfectly with the rest of the structure, as well as to place each electronic component in the proper position. The structure is joined by means of screws placed in the joints between pieces and in areas of greater stress support.

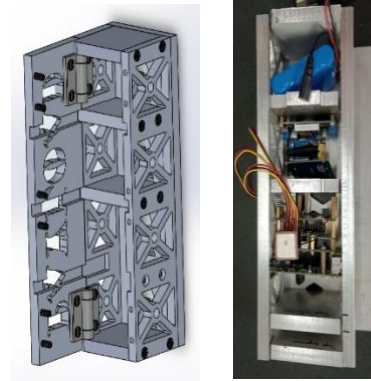


Figure 18. Pre-design 3D modelling of CubeSat Payload case design, after manufacture without and with components.

C. Payload Hardware Design

The schematic diagram of connections within the payload has been designed, where the ports of the Raspberry Pi 4 processor used for the connection of peripherals are visualised. **Error! Reference source not found.** shows the communication type, port and input/output pins used to connect each component to the processing unit.

Table 4. Connections of electronic components with processor

Component	Subsystem	Function	Communication	Port	Outlet Pin (component)	Inlet Pit (processor)
GPS NEO 6M	Tracking and Retrieval System	Global Positioning Sensor	Universal Asynchronous Receiver/Transmitter (UART)	UART5	VCC	3.3V (pin 1)
					TX	TXD5 (pin 32)
					RX	RXD5 (pin 33)
					GND	GND (pin 20)
					5V	5V (pin 4)
Xbee Pro S3B + Xbee USB Adapter	Ground Communications System	Data Transmission Module to Ground Station	Universal Asynchronous Receiver/Transmitter (UART)	UART1	GND	GND (pin 30)
					TX	TXD1 (pin 8)
					RX	RXD1 (pin 10)
					RST	-
Raspberry Pi NoIR Camera V2	Mission System	Photo Capture	Camera Serial Interface	MIPI CSI	-	-
Power Bank	Power System	Main Power Supply	USB	Tipo C	-	-

To ensure the connection between the GPS NEO 6M and the Xbee Pro S3B - Xbee USB Adapter peripherals with the processor, a PCB was designed and fabricated that connects to the Raspberry Pi 4 processor through the GPIO ports (see Figure 20. **Error! Reference source not found.**).

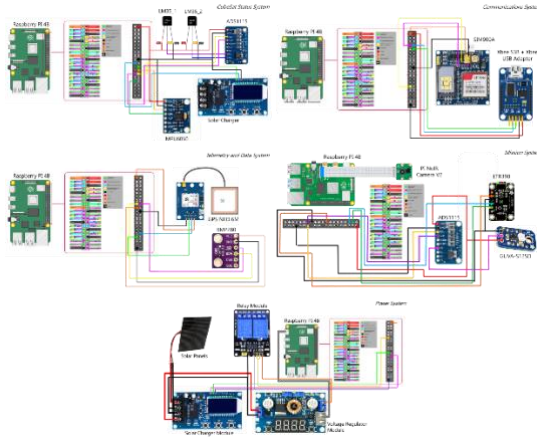


Figure 19. Wiring diagrams of CubeSat subsystems.

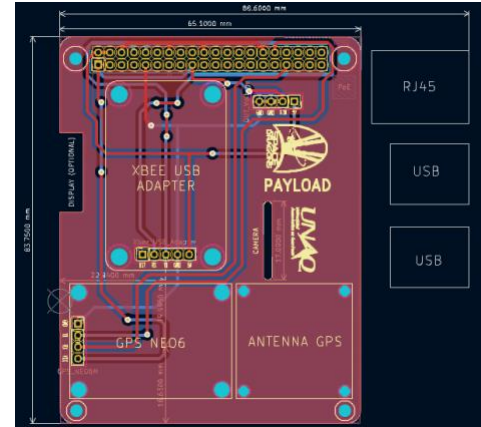


Figure 20. PCB designed to connect peripherals in payload

17. Firmware

The CubeSat is composed of five subsystems that provide power, enable communications between Payload and ground station, as well as allow the fulfillment of the latter's mission of UV radiation level data analysis. Figure 2 shows the data flow diagram between the subsystems and the Raspberry Pi 4B microprocessor used. Figure 19 shows the wiring diagram of each component that makes up the subsystems with the microprocessor.

D. Function tests

Tests have been performed on the following completed subsystems and advances implemented so far operating simultaneously, storing data in the on-board SD memory, and sending data to ground station (view Figures 4).

18. Telemetry and Data System

Acquisition of temperature, pressure, and altitude data from BMP280 sensor by I2C protocol. Obtaining GPS coordinates (latitude and longitude) from NEO6M GPS module by UART serial protocol. It is intended to implement redundant GPS devices in case the NEO6M GPS module loses satellite connection. The SIM900A module of the Communications System and a GPS device independent of the CubeSat subsystems (to be defined) will be used.

19. Mission System

UV radiation intensity acquisition from GUA-S125D sensor by I2C protocol output from ADS1115 analog-to-digital converter. UV radiation index acquisition from LTR390 ALS+UV sensor by I2C protocol.

20. CubeSat Status System

Acquisition of accelerations data in X, Y and Z axes from MPU6050 accelerometer sensor by I2C protocol. Acquisition of battery temperature data by averaging the measurements of two LM35 sensors by I²C protocol output from ADS1115 analog-to-digital converter. Constant operation of fans for Raspberry Pi processor as a cooling system. Obtaining battery charge status data from XY-L10A solar charger module via UART serial protocol.

21. Communications System

Sending data from payload subsystems to ground station via Xbee S3B module with 2.5dBi antenna at 900MHz through UART serial protocol.

Fig. 4. Payload electronics in operation and visualization of subsystem operation on console.

E. Operation time

Together the five CubeSat subsystems connected to the processor consume between 0.5A and 1.5A of current at 5V. The 18650 batteries used provide 7.4V and a maximum of 2200mAh. Based on the above we can estimate that the batteries will last X hours with all the electronic components of each subsystem working simultaneously. We intend to implement a power system capable of recharging the batteries by means of solar panels once the duration time has elapsed to guarantee the operation of the CubeSat from the pre-launch stage until its recovery.

V. Ground Station, Telemetry On-Board Computer and Dual Deployment Altimeter Subsystems

The telemetry on-board computer serves as the electronic hub of the system, receiving inputs from various sensors to measure altitude, pressure, location, and acceleration changes. This information is sent via RF (LoRA Frequency Band 902-928 Mhz.) to the ground station to display and log the flight data. To increase vehicle location redundancy, an independent backup GPS subsystem has been added. The system consists of a transmitter and a handheld receiver developed by Apogee Components. The backup GPS also transmits location data via RF in the LoRA Frequency Band. Finally, the dual deployment altimeter subsystem consists of two redundant COTS altimeters RRC3 developed by Missile Works, which determine the optimal time to deploy the drogue parachute at apogee, estimated to occur around 10,000 ft AGL. Then, once the rocket has reached the target altitude of 1,000 ft AGL, the main parachute is deployed to further reduce the rate of descent and bring the rocket safely to the ground. In the event of a failure of the main deployment altimeter, an additional redundant deployment altimeter has been implemented to ensure that the parachutes can still be deployed. The main functions carried out by each subsystem consist of:

Dual Deployment Altimeters

- Activate the two-stage recovery system of the dual deployment mechanism.
- Log the flight data into the flash memory.

Telemetry On-board Computer

- Transmit real-time GPS coordinates, altitude and acceleration of the vehicle to the ground station.

Ground Station

- Receive data from Telemetry On-board computer and Payload, display it using a local web application and log data to a database.

Backup GPS

- Transmit real-time GPS coordinates to handheld device.

The decision has been made to harness commercial components that have undergone testing within the industry. This decision was made to ensure the safety of this critical system. The design specifications include the form factor, estimated weight, and materials used:

- Form Factor: Cylinder Estimated Weight: 3.3 lb.
- Material: ABS filament and aluminium
- Base Diametral Dimension: 5 in
- Height: 16.22 in.

The rocket structure securely bolts the cylinder-type avionics bay using four threaded bolts to ensure robust attachment and balanced weight distribution. This configuration also provides a stable and secure platform to hold critical electronics during flight. Two aluminium discs absorb the pressure impact in their respective compartments during parachute deployment.

For the Telemetry On-Board Subsystem, a PCB was designed to reduce wiring and increase reliability between the different sensors and a microcontroller Arduino Nano.

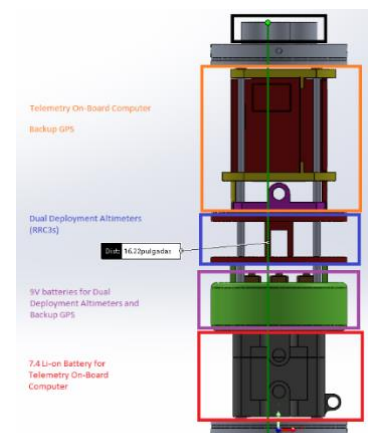


Figure 21. Rendering of proposal for electronics bay

The telemetry on-board computer includes an inertial measurement unit that combines an accelerometer and a gyroscope on a single chip (MPU6050 with I2C protocol), a low power consumption pressure and temperature measurement device that is also highly accurate (BMP280 with SPI protocol) to measure altitude, a voltage regulator (LM2596), and a GPS NEO module (UART protocol).

Furthermore, an XBee S3B Pro with an RP-SMA antenna will be used to transmit data in real time. The power supply system will be provided by four cylindrical 18650 lithium-ion metal case batteries. The battery pack is wrapped with thermofit plastic and provides 7.4v with 4,400 mAh capacity. The following image shows the schematic connections of the telemetry system.

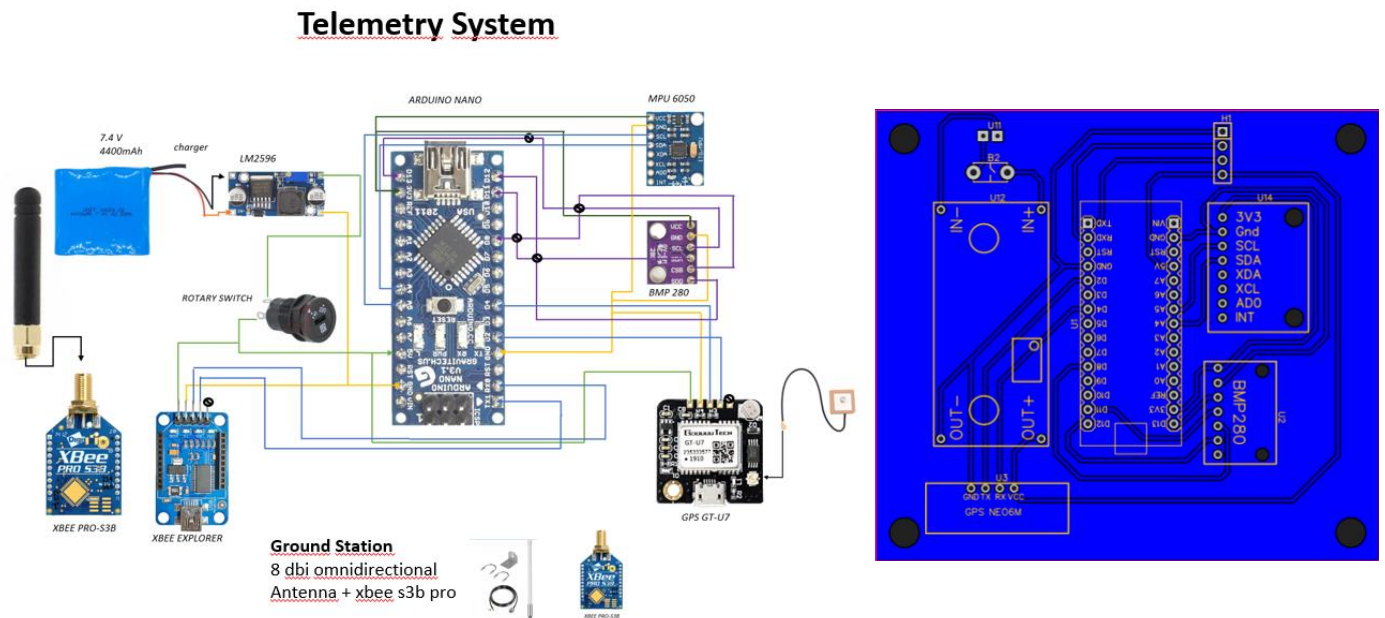


Image 1: PCB Design for telemetry system

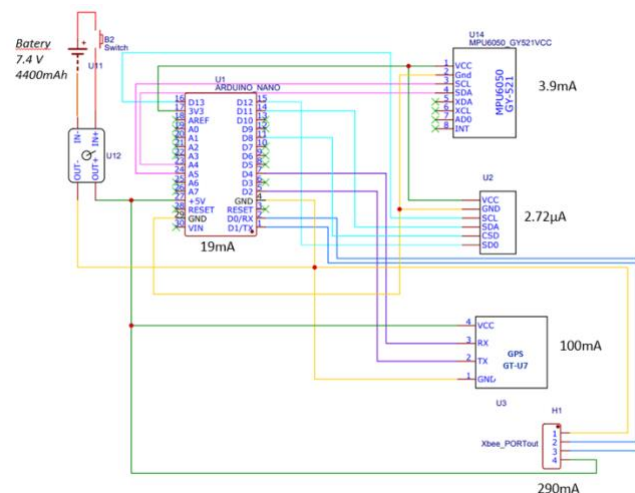


Figure 22. Telemetry system schematics

Table 5. Key information of the subsystems

Subsystem	Current Draw (mAh)	Estimated Battery Life (hours)	Range (miles)	Frequency Band (Mhz)	Switch Type	Power Source	High Gain Antenna Receiver	Wiring
Telemetry	415.62	10.58	9 Miles	902-928	Rotary	7.4v Lion 4400 mAh.	Omnidirectional - 8dbi	PCB, 26 AWG for Xbee. 20 AWG power.
Dual Deployment Altimeters	35.00	13.00	N.A.	N.A.	2 x Independent Rotary	2 x independent 9V 455 mAh alkaline batteries.	N.A.	20 AWG.
Backup GPS	180.00	7.22	30 Miles	908-928	Rotary	9v Lion 1300 mAh.	Yagi - 8dbi	20 AWG.

VI. Test Flight Data

Sunday April 19th was undertaken the team test flight on the Tripoli México launch site Laguna Salada, Mexicali, Baja California on coordinates 32.33716225278786, -115.67123182447533. The rocket was lunched with an Aerotech M685W-PS rocket motor reaching an apogee of 8,447 ft, a maximum velocity of 635 ft/s, an ascent time of 25.05 s until apogee and a descent time of 148.75 s.

After our test flight, the data log was recovered from both RRC3s. The following images show the graphs from the data recovered.

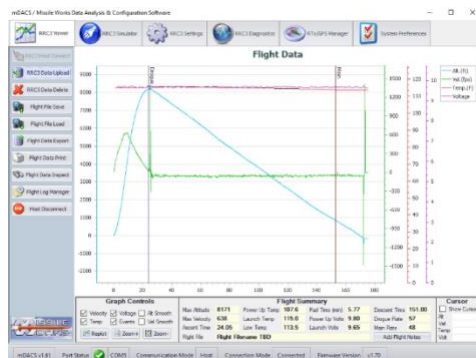


Figure 24. Primary Altimeter

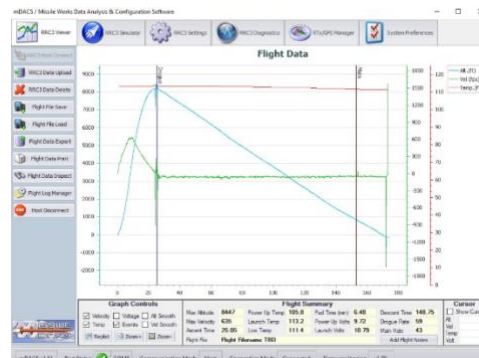


Figure 25. Secondary Altimeter

As indicated in the graphs, the vehicle reached an altitude of 8,171 ft while the second deployment altimeter recorded a slightly higher altitude of 8,447 ft. Additionally, both deployment altimeters reported a velocity of approximately 635 ft per second. The main conclusion drawn from this data is that the rocket fell short of the 10,000 ft. AGL. Consequently, measures to reduce weight were being explored to increase the target apogee.

A. Software simulations

22. Test Flight RASAero II

For these simulations we consider the launch site conditions to emulate the ones we had on mid-April on the Laguna Salada, a launch site elevation of 0 ft, a temperature of 88 °F, a barometric pressure of 29.91 in/hg a windspeed of 13 mph, a launch rail length of 12 ft and a launch angle form vertical of 4°; in addition it was simulated with a AeroTech M685W-PS motor, considering a weight of 59 lb and 4 rail buttons on the aft and top airframe as shown.

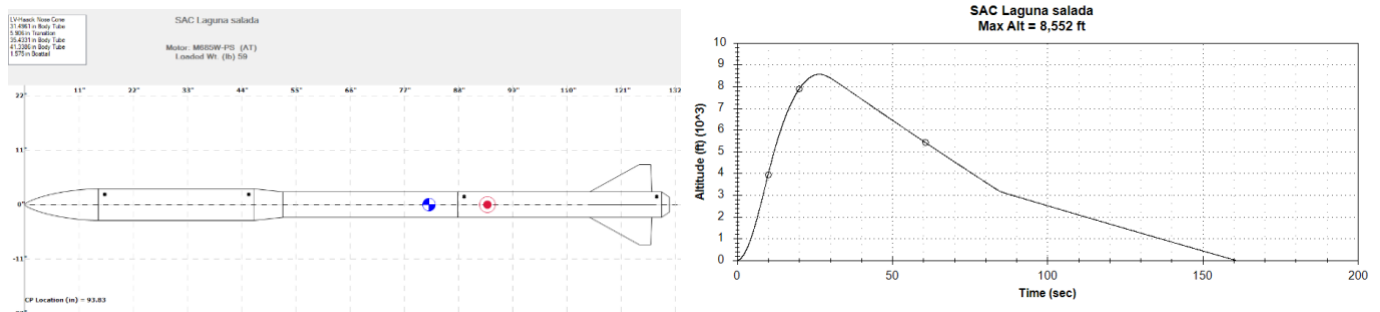


Figure 26: Analysis obtained from the test flight

23. Spaceport Contest RASAero II

For these simulations we consider the next launch site conditions to emulate the ones we will have on mid June on Spaceport America Cup, a launch site elevation of 5692 ft, a temperature of 95 °F, a barometric pressure of 29.73 in/hg a windspeed of 16 mph, a launch rail length of 17 ft and a launch angle form vertical of 4°; in addition it was simulated with a AeroTech M1850W-PS motor, considering a weight of 55 lb and 2 rail buttons on the aft airframe as shown

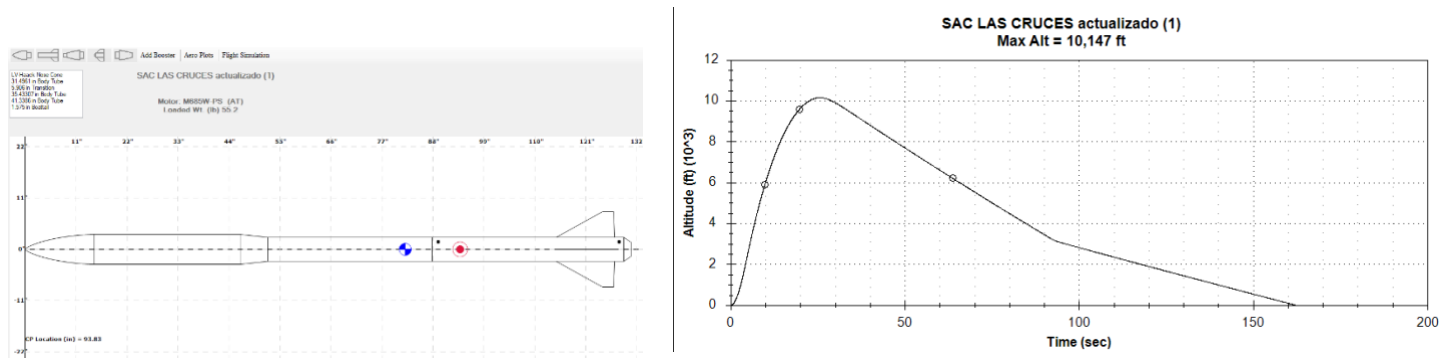


Figure 27: Analysis obtained from the future launch at Las Cruces

VII. Mission Concept of Operations Overview

The prelaunch phase starts when the launch vehicle is raised vertically on the launch rail, and final preflight preparations begin. During this phase, the ground station confirms the reception of data from both flight computer and payload. Additionally, the electronic arrangement for the electronic bay is turned on through rotatory switches confirming the main data systems are working correctly. After reviewing and completing the pre-flight checklist in the Assembly, Pre-flight and Launch Checklists Appendix, the engine ignition signal initiates the ignition phase, which triggers the ignition of the electrical matches and the engine. During the liftoff phase, the project accelerates along the launch rail and exits with a velocity of 55 ft/s and a stability calibre of 2.08. During this phase, the rocket will be in flight for 28.70 seconds before reaching its predicted maximum altitude. The engine will continue to burn for approximately 5 seconds, followed by an inertial ascent to an altitude.

The deployment of the drogue parachute begins when the main dual deployment altimeter detects that the vehicle has reached apogee and trigger the e-match in the respective black powder container. The lower airframe A separates from the rest of the vehicle and begins to descend under its own parachute, dropping at a speed of 77.099 ft/s. If the first powder charge fails to activate, the redundant system will detect the completion of the 1-second apogee delay and activate the second deployment burst charge.

The main recovery phase begins when the altimeter records an altitude of 1,000 ft AGL. At this point, the recovery altimeter recognizes that the previously established altitude has been reached, which triggers the second deployment event. During this event, the second explosive charge is activated, and the mechanism for deploying the main parachute is released, reducing the rate of descent to 29.363 ft/s. To ensure the CubeSat's integrity, the payload will descend at a speed of 45 ft/s.

Finally, both the rocket and payload descend independently, and the recovery and deployment phase conclude once they have both landed. The final phase is search and recovery.

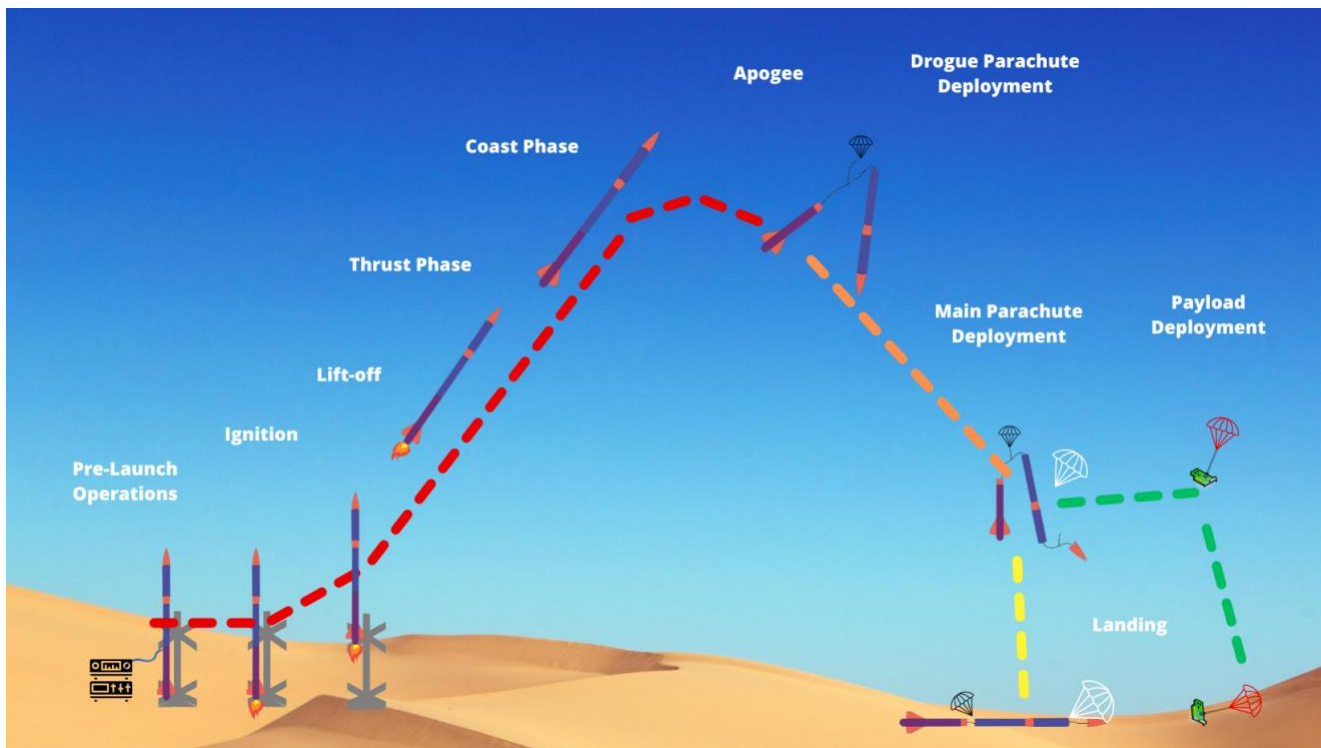


Figure 29. Concept of Operations Overview. The orange path indicates phases that are mutual with the rocket and with the payload, green path tracks the vehicle's activities after apogee, and gray path tracks the payload's activities after deployment.

VIII. Conclusions

During this technical report, the team has established the parameters for each geometry during development while manufacturing of the project will begin in early 2024. However, valuable lessons were learned during the development of the first deliverable. During this initial project period, the focus was on designing and conceptualising the rocket vehicle and mission. The members have gained valuable experience through our international participation in Spaceport America Cup 2023. This experience provided a better understanding of rocketry and the challenges the team will face. Nevertheless, material selection has been a constant source of uncertainty in this project because of the team's limitations. The objective was to innovate by proposing manufacturing based on composite materials and machining, rather than additive manufacturing just as 3D printing. For the next deliverables, the level of demand and available time for manufacturing each component will be important considerations. We are ready to contest with high grade of excellence this year.

IX. Appendix on Weight Systems, Measurements and Performance Data

A. General Project Information

- Number of Stages: 1
- Airframe Diameter: 5 in and 6 in
- Fins number: 4
- Rocket Length: 134.646 in
- Fin semi-span: 4.724
- Tip chord: 2.756 in
- Root chord: 12.598 in
- Thickness: 0.8 in
- Total payload and vehicle weight (without motor): 30.542 lb
- Total payload and vehicle weight (with motor): 46 lb
- Payload Weight: 8.9 lb
- Centro de presión: 95.275 in
- Centro de gravedad: 82.677 in

Propulsion Overview

- Motor: Commercial
- Manufacturer: AeroTech
- Classification: M1850W-PS
- Average Thrust: 1,909.6 N
- Total Impulse: 7,658 Ns
- Thrust Duration: 4 seconds

Flight Data

- Take-off thrust-to-weight ratio: 1.0 to 8.7
- Exit velocity from launch rail: 55 ft/s
- Maximum acceleration: 186.680 ft/s²
- Maximum velocity: 738.189 ft/s
- Predicted apogee: 1010.400 ft

Payload Overview

- Form Factor: 3U
- Estimated Total Weight: 8.9 lb
- Armour Weight: 7.606 lb
- Electronic System Weight: 1.294 lb
- Base Dimension: 10 cm × 10 cm
- Height: 30 cm

X. Appendix of Project Test Reports

A. Deployment test

24. Nose cone and aircraft.

For both the ejection of the nose cone and the fuselage only the main charge was tested, and the procedure was as follows:

1. Clean the cavity where the charge of BP and the electronic match will be placed.
2. Insert the electronic match through the hole on the disk leaving the ignited inside of the cavity.
3. Add the calculated amount of BP to the cavity (5g for the drogue and 4g for the main parachute).
4. Secure the electronic match and BP charge with masking tape on the top of the cavity.
5. Place the main computer in its position on the vehicle.
6. Assemble the parts that will be ejected and secure them with shear pins.
7. Activate the electronic match with a battery.

The evidence for both tests are presented below:



Figure 301. Deployment of the lower fuselage and parachute due to the explosive charge.



Figure 31. Explosion of the drogue charge in ground test.

Both attempts, the main parachute and the drogue parachute were successful on the ground test, indicating that the calculated amount of BP was sufficient.

B. Flight test

The flight test took place on April 19th on the Tripoli México launch site Laguna Salada, Mexicali, Baja California on coordinates 32.33716225278786, -115.67123182447533, it was tested with an Aerotech M685W-PS rocket motor, where it reached an apogee of 8447 ft, a maximum velocity of 635 ft/s, an ascent time of 25.05 s until apogee and a descent time of 148.75 s.



Figure 32. Rocket at launch test

C. Computer test

As part of the tests, we also did a computer test to see if the barometer works, we used a SRAD vacuum system to test the detonation for barometric sensors and another one to test the detonation while connected to the ground station.



Figure 33. Vacuum altimeter test

XI. Appendix of Risk Evaluation

Space Dragons Rocketry Team				
Danger	Possible causes	Risk level	Mitigation	Risk level after mitigation
Motor ignition during the transfer to the launch site	Placing the motor near a heat source	High; severe damage to the transport vehicle	Place the motor in a thermally insulated container	Low
Black powder explosion during installation of the Recovery System	The Black powder is placed near the batteries of the electronic recovery system and generates a spark	High; Damage to launch crew, recovery system and facilities	Clear the work area of any electrical source	Low
Physical Exposure to Fiberglass dust while working on the airframe	Lack of personal protection equipment	High; Skin irritation of launch personnel	Wear lab coat and personal protection equipment	Low
Inhalation of fiberglass dust when sanding and cutting the airframe.	Lack of nasal protection when performing these activities	High; Damage to the health of the launch crew	Use a face mask with smoke and dust filters	Low
Short circuit during main electronic computer connection	Errors during electronics connection or a misplaced connection	High; jeopardizes the deployment and recovery system.	Checking connections with a multimeter and using redundancy systems	Low
Avionics failure	Connections are mispositioned, batteries are damaged or without power	Medium; Insufficient time to install and configure the avionics	Inspect the system and its connections using the checklist	Low
Misfire with the e-match and ignitors	Its operation was not verified	Medium; don't work as it suppose	The necessary tests have not been carried out	Low
The fins are not firmly attached to the airframe	Lack of epoxy resin in fillets	Medium; No pre-verification has been performed	Apply Instant Epoxy Resin	Low
Solid propellant rocket motor explosion	Motor casing cannot contain operating pressure due to manufacturing defect	Medium; Necessary inspections were not carried out	Perform casing and assembly inspections using its respective checklist	Low
The altimeters of the main computer did not detect the set parameter	Altimeter decalibration	Low: A calibration plan was not followed	Inspect the system prior to assembly and ensure calibration of the altimeter	Low

XII. Appendix of Assembly, Pre-Flight and Launch Checklist

A. Preflight checklist

X	Preparation of conditions
	Walk towards the launch platform and rail.
	Lower the rail to a horizontal position.
	Slide the rocket buttons over the rail.
	Check the correct coupling of the rocket.
	Raise the rail to its vertical position and fix.
	Insert two lighters under the engine nozzle until reaching the top.
	Secure the lighters with adhesive tape.
	Discharge the crocodile clips by touching them together.
	Assemble the lighters, peel the lighter cables and wrap several times around the crocodile clips.
	Separate the clips at least 5 inches to avoid contact with each other.
X	Conditions of the electronics of arming
	Place the ladder next to the rocket.
	Start the arming sequence when the rocket is placed vertically on the launch platform.
	Turn on the backup GPS switch.
	Wait until Satellite Lock is indicated on both the portable receiver and the transmitter. The coordinates and direction of the device should be seen.
	Turn on the telemetry switch.
	Wait for confirmation from the ground station of data reception.
	Turn on the main switch of the flight computer.
	Listen for a 5-second beep.
	Wait for the altimeter calibration and confirm 3 sequential beeps.
	Turn on the redundant switch of the flight computer.
	Listen for a 5-second beep on the second computer.
	Wait for the computer's altimeter calibration to start 3 sequential beeps.
	Communicate with the receiving station and confirm that the GPS signals are acquired and functioning properly.

Prelaunch checklis

X	Launch protocol
	Verify weather conditions.
	Verify that the sky is clear, and the wind speed is below the limits.
	Assemble the switch that energizes the system.
	Start the countdown beginning the count at 10.
	Press the launch button that ignites the rocket engine.

Recovery checklist

X	Recovery conditions
	Ensure that the recovery team wears safety equipment (boots) and has 4 hours, worth of water, sunscreen, insect repellent, had sunglasses, long sleeves and pants, GPS and radio.
	After 4 hours of searching return to the recovery tent and request another deployment process.
	Check tracking frequencies in the Tracker check tent before flight.
	After the launch, collect the last location received by the GPS.

	Verify radio communication, GPS tracking and real-time mapping.
	Check radio every 30 minutes from start of search.
	Once you have found the rocket, do not touch or move any components.
	Be sure to carry 3 to 4 gallons of water to stay hydrated.
	Protect extremities and limbs with the necessary equipment, as well as wear glasses and appropriate footwear.
	Locate the rocket within the established landing zone.
	Once all rocket parts and payload have been recovered, return.
	Return the Communication and Tracking Package to the Mission Control Center.
	Present the rocket for post-launch evaluation.

Off nominal checklist

X	Conditions for disassembly.
	In case of engine failure, do not approach the launch platform for at least 5 minutes.
	Perform a visual inspection from a safe distance to verify that there are no visible signs of smoke or fire at the nozzle.
	Use personal protective equipment before approaching the rocket.
	Turn off the rocket ignition system.
	Disconnect the alligator clips from the ignition system.
	Disconnect the switch from the flight computers.
	Disconnect the switch from the redundant GPS and telemetry.
	Unload the engine and inspect if the ignition system has started.
	Replace with a new ignition system, following the checklist and the appropriate procedure for assembling the engine.
X	Flight computer failure.
	Complete the disarmament procedure checklist.
	Extract the electronic compartment.
	Inspect the electrical connections to the power supply and power switches.
	Measure the output of the power supply and verify if the outputs are within the specified parameters.
	If anomalies are detected, change the power supply.
	Check if there are short circuits in the electronic bay.
	If the failure continues, perform a continuity check on the circuit board.
	In case of breakdown, replace the components.
	Perform the proper coupling of the electronic bay in the rocket.

XIII. Appendix of Engineering Drawings

The following pages contain the engineering drawings generated so for the SD Rocket II to be presented at SAC 2024.

1.005	
Carbon fiber & Aluminum	Inner tube & center ring
1	
	Space Dragons 10/05/2024
1	5
	5